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MEMORANDUM

A WEIGHT COMPARISON OF SEVERAL ATTITUDE CONTROLS FOR SATELLITES

By James J. Adams and Robert G. Chilton

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Langley Field, Va.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

WASHINGTON

February 1959

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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SUMMARY

A brief theoretical study has been made for the purpose of estimating and comparing the weight of three different types of controls that can be used to change the attitude of a satellite. The three types of controls are jet reaction, inertia wheel, and a magnetic bar which interacts with the magnetic field of the earth. An idealized task which imposed severe requirements on the angular motion of the satellite was used as the basis for comparison.

The results showed that a control for one axis can be devised which will weigh less than 1 percent of the total weight of the satellite. The inertia-wheel system offers weight-saving possibilities if a large number of cycles of operation are required, whereas the jet system would be preferred if a limited number of cycles are required. The magnetic-bar control requires such a large magnet that it is impractical for the example application but might be of value for supplying small trimming moments about certain axes.

INTRODUCTION

Earth satellites offer unique opportunities for obtaining scientific data. Although much information can be obtained from unstabilized satellites, many instrumentation applications require a stabilized reference attitude with respect to the earth or the sun. The weight of the energy-storing and torque-producing devices required to control the attitude of a satellite is an important consideration in the choice of a control. Therefore, it might be asked whether certain basic methods of control may have outstanding weight-saving advantages over others. A brief study has been made for the purpose of estimating and comparing the weight of three types of control about a single axis. The three types of controls are jet reaction, an inertia wheel, and a magnetic bar which interacts with the magnetic field of the earth.

In the study of this paper, the general expressions for items that will affect the weight of the system such as the weight of fuel required,

energy required, or power required are described. From these general expressions, the weight of the torque-producing part of the control can be estimated. Even when the choice of a control is based on weight alone, a different choice may result for every application depending upon the size of the satellite, the operational life, and the stabilization requirements. A specific example is used in this paper as a basis for comparison for the three types of controls. The task used is an idealized one and was chosen because it imposed a rather severe requirement on the angular motion of the satellite.

SYMBOLS

A	area, sq ft
a,b	constants
B	magnetic field intensity, weber/meter^2
c	radius of earth, ft
E	energy, ft-lb
g	constant of gravity, ft/sec^2
h	altitude of orbit, ft
I	moment of inertia, slug-ft^2
I'	magnetic intensity, weber/meter^2
I_{sp}	specific impulse, sec
l	length, ft
P	power, ft-lb/sec
R	tracking line, ft
r	radius of orbit, ft
T	torque, ft-lb
t	time, sec
V	velocity, ft/sec

ω	angular velocity, radians/sec
$\dot{\omega}$	angular acceleration, radians/sec^2
$\epsilon,\beta,\alpha,\theta,$ ψ,ϕ,δ	angles, radians (fig. 1)
μ_o	permeability of a vacuum

Subscripts:

S	satellite
R	rotor or flywheel
T	target
E	earth
max	maximum

Bars over symbols indicate vectors.

CONTROL TASK

The details of the specific task used to compare the control systems are as follows. The satellite is assumed to be a 3,000-pound object with a moment of inertia about the control axis of 1,000 slug-feet^2. Such an object might be a sphere with a diameter of 10 feet and with a uniform density. The required mission is to track continuously a ground target from an orbit altitude of 300 miles as it appears on the horizon, passes beneath, and disappears over the opposite horizon. (See fig. 1.) It is assumed that the satellite S is initially pointing at the target T when it appears on the horizon. The task is repeated for each cycle of the satellite orbit for an indefinitely large number of cycles. This arbitrary specific task is used to illustrate the application of the general considerations involved in determining the weight of the control system. The example chosen is useful in that it combines high energy and power requirements. Also, if a relatively low altitude is chosen for the orbit, severe angular motions are stipulated. In order to make the results more general, a brief discussion of the effect of changing the attitude time history will be given. Other time histories may emphasize either power or energy requirements.

The geometry of the assumed tracking task is shown in figure 1. The vector expression for the angular velocity of the tracking line R, which

is equivalent to the total angular velocity of the satellite about its own axis, is given by the equation:

$$\bar{\omega}_S = \frac{\bar{R} \times \left(\bar{V}_T - \bar{V}_S\right)}{R^2} \tag{1}$$

Assume that the orbit is circular and lies in the equatorial plane. Then equation (1) reduces to

$$\omega_S = \frac{-r\omega_{S_1}}{R}\left(\sin \alpha - \frac{c\omega_E}{r\omega_{S_1}} \sin \epsilon\right) \tag{2}$$

where ω_E is earth's angular velocity and ω_{S_1} is the angular velocity of the satellite about the earth. The effect of the earth's rotation appears in the second term within the parenthesis in equation (2). The effect is greatest for the equatorial orbit. For the minimum-altitude orbit the earth-rotation term makes a maximum contribution of about 6 percent to the magnitude of the angular velocity of the tracking line. For the purpose of this study the effect of the earth's rotation will be neglected and the angular velocity of the tracking line expressed in equation (2) reduces to

$$\omega_S = -\frac{r\omega_{S_1} \sin \alpha}{R} \tag{3}$$

For a circular orbit the angular velocity ω_{S_1} of the satellite is a constant given by

$$\omega_{S_1}{}^2 = \left(\frac{d\theta}{dt}\right)^2 = \frac{b^2}{(1 + a)^3}$$

where

$$b^2 = \frac{g}{c}$$

$$a = \frac{h}{c}$$

From geometrical considerations of the circular orbit,

$$\sin \alpha = \cos \beta = \frac{r - c \cos \theta}{R}$$

and

$$R^2 = c^2 + r^2 - 2rc \cos \theta$$

where

$$r = c + h$$

Combining these expressions results in

$$\omega_S = -\frac{b}{2(1 + a)^{3/2}} \times \frac{(1 + a) - \cos \theta}{1 + \frac{a^2}{2(1 + a)} - \cos \theta} \tag{4}$$

The angular acceleration of the tracking line is obtained by differentiation of equation (4) as

$$\dot{\omega}_S = \left(\frac{d\omega_S}{d\theta}\right)\left(\frac{d\theta}{dt}\right) = \frac{ab^2(2 + a)\sin \theta}{4(1 + a)^4\left[1 + \frac{a^2}{2(1 + a)} - \cos \theta\right]^2} \tag{5}$$

The angular velocity and angular acceleration for a 300-mile-high orbit are plotted in figure 2. The maximum angular velocity required occurs as the satellite passes over the target and is less than 1° per second. The maximum control torque, which is determined by the product of the peak angular acceleration and moment of inertia, is 0.14 foot-pound for the assumed satellite.

Figure 3 is a plot of the variation of the magnitude of the product of angular acceleration and angular velocity with time. The power per unit inertia required to maneuver the satellite in the prescribed manner is a function of this product. The peak value is 1.6×10^{-6} foot-pound per sec per slug-foot^2. The energy absorbed is a function of the area under the curve, which is 2.2×10^{-4} foot-pound per cycle per slug-foot^2 or 3×10^{-4} watt-second per cycle per slug-foot^2.

CALCULATIONS

The purpose of this study is to determine whether certain basic types of attitude control may have outstanding weight-saving advantages over others. Three basic types of control were chosen for comparison; namely, jet reaction, inertia wheel, and magnetic bar. Admittedly, within each of the three categories, the design for the application of the basic

principle may vary widely in detail. The system configuration chosen to represent each basic type of control may not be the most efficient possible; however, significant differences in the weight of the systems are expected to be revealed by the study.

In the following sections each attitude control system is described and an estimate is made of the total weight of the control system plus the energy source.

Jet Reaction

The use of jets seems a logical choice for satellite control as does the use of hydrogen peroxide for fuel, because of the simplicity of using a monopropellant and because of its self-starting characteristics. The two items in the jet control that are related directly to the requirements of the specified maneuver are the weight of the jet-nozzle assembly and the weight of the fuel. The weight of the nozzle is related to the maximum angular acceleration required but is not related to the maximum angular velocity. However, a more important item in the weight of the system is the amount of fuel required, which is a function of the angular velocity required and the number of cycles of operation. When the specific impulse of hydrogen peroxide is used, the amount of fuel required can be determined as follows.

The maximum theoretical specific impulse of hydrogen peroxide is approximately 190 pounds of thrust per pound of fuel per second. However, the maximum specific impulse obtained in tests at sea-level atmospheric conditions is 125 to 130 seconds. The value that can be realized should increase as the condition of a vacuum is approached. The thrust required is related to the motion of the satellite as follows:

$$\text{Torque} = \text{Thrust} \times l = I_S \dot{\omega}_S$$

where l is the distance between jets. An expression for the fuel rate can now be written as

$$\text{Fuel rate} = \frac{\text{Thrust}}{I_{sp}} = \frac{I_S \dot{\omega}_S}{l I_{sp}}$$

By integrating this expression the fuel required for the given task can be determined

$$\text{Fuel required} = \frac{I_S}{l I_{sp}} \int_0^t \dot{\omega}_S \, dt$$

Since the angular velocity for the assumed task is symmetrical about the half cycle and is zero at the start and finish of the cycle, the integral $\dot{\omega}_S$ dt over the time of the cycle can be expressed as

$$\int_0^t \dot{\omega}_S \, dt = 2\omega_{max}$$

where ω_{max} is the maximum angular velocity of the satellite. Note that the weight of fuel required can be reduced by increasing the moment arm l. The extent to which it would be practical to provide this increase would depend on the weight of the structure necessary to extend the arm and the number of cycles of the maneuver that are required.

If the value of 130 seconds is taken for the specific impulse and the distance between the jets is taken as 10 feet in order to make a conservative estimate, the weight of fuel required per cycle for the assumed configuration is 0.023 pound. The minimum weight for the total system, including the weight of four jets, supply lines, pump or pressure tank used to deliver the fuel to the jets, and supply tank is estimated to be 6 pounds. The weight of the supply tank was assumed to increase as the weight of fuel required increases.

Inertia Wheel

The second type of control investigated was an inertia wheel. In this case, an electric motor rotor or rotor-flywheel combination accelerated in one direction causes the satellite to accelerate in the opposite direction. In the absence of any external disturbing torques, one of the relationships that must be satisfied is that the angular momentum of the satellite must equal the angular momentum of the rotor

$$I_S\omega_S = I_R\omega_R$$

Thus it can be seen that, for a given satellite configuration, the maximum angular velocity of the satellite is related to the maximum angular momentum of the rotor. Also the angular acceleration of the satellite is related to the torque output of the motor

$$I_S\dot{\omega}_S = T = I_R\dot{\omega}_R$$

The most significant point insofar as the weight of the system is concerned is that the moment of inertia of the rotor should be large. The twofold reason for this requirement is explained as follows. The power out-put of the motor is defined as

$$P = T\omega_R = I_R\dot{\omega}_R\omega_R$$

or, for convenience,

$$P = I_S\left(\frac{I_R}{I_S}\right)\dot{\omega}_S\left(\frac{I_S}{I_R}\right)\omega_S\left(\frac{I_S}{I_R}\right) = I_S\dot{\omega}_S\omega_S\,\frac{I_S}{I_R}$$

Thus it can be seen that the maximum power output of the motor is proportional to the ratio of the moment of inertia of the satellite to the moment of inertia of the rotor. Since the weight of the motor will be a function of its power output, it is desirable that the moment of inertia of the rotor be large.

The second factor in the weight of the system that is affected by the size of the rotor is the weight of the energy storage device. The energy required for one cycle of the maneuver is the integral of the power over the time of the maneuver.

$$E = \int_0^t P\,dt = I_S\left(\frac{I_S}{I_R}\right)\int_0^t \dot{\omega}_S\omega_S\,dt$$

It can be seen that the energy required, like the power required, is a function of the ratio of the moment of inertia of the satellite to the moment of inertia of the rotor and the desired motion of the satellite. Suppose a 5-pound flywheel which has a thin web and a thick rim with a diameter of 1 foot is attached to the rotor. The moment of inertia of the flywheel is 0.039 slug-ft^2. Therefore, the required maximum power output of the motor will be approximately 1/10 horsepower for the task considered here. A survey of the commercially available electric motors indicates that a direct-current motor with this output weighs approximately 4 pounds. The total energy output of the motor for one cycle is 5,650 foot-pounds. A zinc-silver battery, which represents an efficient and available type of battery, may be used to store this energy. Reference 1 states that a zinc-silver battery will store 146,000 foot-pounds per pound of battery. Therefore, a 0.039-pound battery will be required for the useful work per cycle.

Provision will also have to be made for the energy losses that occur in the motor. These losses are the result of friction and heat and vary with the power-output level of the motor. The efficiency of the motor varies from approximately 80 percent at rated output to 0 percent for a no-load constant-angular-velocity condition. As an estimate, it is assumed that the average efficiency of the motor during a cycle is 60 percent. Therefore, 0.065 pound* of battery would be required for every cycle of the example maneuver.

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*Some of the energy stored in the flywheel could be returned to the battery during the decelerating part of the cycle by making the motor act as a generator so that the amount of energy that would have to be stored and, consequently, the required weight of the battery could be reduced. However, no consideration of this factor was included in this analysis.

The minimum weight for this flywheel control is estimated to consist of a 5-pound flywheel, a 4-pound motor, and a minimum size 2-pound battery for a total of 11 pounds. This minimum battery would be sufficient for 30 cycles.

There are two factors that will improve the inertia-wheel control from a weight standpoint. The first of these is to add part of the satellite equipment to the flywheel. This will increase the moment of inertia of the flywheel without increasing the weight penalty. An extreme example of this variation is to divide the satellite into two separate parts with equal inertia and have the motor supply a torque to their common axis. Assume that each section has a moment of inertia of 500 slug-feet^2. The power and energy required are functions of this inertia and the relative angular velocity and acceleration of the two sections. The angular velocity and acceleration are twice the calculated required values for the satellite rather than equivalent to the angular motion of the rotor alone that was used in the previous case. In the previous case the motion of the satellite was so small with respect to the motion of the rotor that it was neglected. The ratio I_S/I_R is unity for the present case. The maximum power output is computed to be 3.2×10^{-3} foot-pounds per second or 5.8×10^{-6} horsepower. A very small motor plus a reduction gear probably similar to those of an electric alarm clock would be sufficient. This power is so small that the energy required would probably be a function primarily of the friction in the drive mechanism. Even if the mechanism were only 1 percent efficient, it can be shown that a 2-pound battery, such as was assumed as a minimum size in the previous case, would be sufficient for several thousand cycles. The total weight of the control, which will consist of the battery and motor only, need not exceed 4 or 5 pounds. Of course, the satellite configuration necessary to achieve this small weight may be impractical because of the difficulty in applying control about more than one axis. The estimated weight is of interest in that it shows the reduction in weight that is possible with the inertia-wheel system.

Another means which can be used to reduce the weight of the system, if a large number of cycles is planned, is to use solar batteries. Solar batteries weigh 0.3 of a pound per watt output if exposed to the sun full time, or 0.8 of a pound per watt if exposed to the sun 50 percent of the time. If it is assumed that the orbiting period for the satellite is $1\frac{1}{2}$ hours and that the solar batteries can be exposed 50 percent of this time, then 2 pounds of solar batteries will supply the energy required for the maneuver when the 5-pound flywheel is used. Of course, a minimum-size storage battery will have to be used to store the energy until the maneuver takes place. The minimum weight of the control would then consist of a 5-pound flywheel, a 4-pound motor, a 2-pound storage battery, and a 2-pound solar battery for a total weight of 13 pounds. This configuration would be sufficient for an indefinite number of cycles.

The inertia-wheel control system is subject to one fundamental consideration. If there exists a constant moment that the system must trim, the motor will eventually be required to exceed its maximum angular velocity. This situation will be discussed further in a later section.

Magnetic Bar

The third system considered is a magnetic bar similar to a large compass needle. A permanent magnet is used to supply the required torque by placing it at some angle to the magnetic field of the earth. Such a system could be used to supply yawing and rolling moments on an equatorial orbit, yawing and pitching moments on a polar orbit at the equator, and pitching and rolling moments at the magnetic poles. The strength of the magnetic field of the earth is given in reference 2. The variation in the strength of the magnetic field with altitude is given by the equation

$$B_{\text{Altitude}}^{-2} = B_{\text{Sea level}}^{-2}\left(1 + 6X + 15X^2\right)$$

where $X = \left(0.478 \times 10^{-7}\right)h$.

The expression for the magnetic moment exerted on a bar magnet of length l with a cross-sectional area A when placed in a uniform magnetic field of strength B at an angle ψ with respect to the direction of the field is given by the equation

$$T = \frac{B}{\mu_o} I'Al \sin \psi$$

where T is the moment, I' is the magnetic intensity of the magnet, and μ_o is the permeability of free space and the units are in the mks system.

The maximum torque is obtained when the magnet is oriented 90° with respect to the magnetic field. The weight of the magnet is estimated by solving for the volume which will give the maximum required control torque. The assumed values of the parameters are as follows: The earth's magnetic field intensity at the equator is 0.31×10^{-4} webers per square meter. The corresponding value of B at the altitude of the orbit is 0.25×10^{-4} weber per square meter. The permeability μ_o is 12.7×10^{-7} in mks units. The maximum magnetic intensity for Alnico V material is 1.6 webers per square meter and the residual intensity is 1.2 webers per square meter. By using the value 1.2 for the magnetic intensity and expressing the required torque in newton meters, the required volume of the bar can be obtained in cubic meters. Expressed in more convenient units the result of the above calculation is a volume of approximately

600 cubic inches or a weight of 180 pounds. This large weight makes the magnetic-bar control an impractical means for obtaining the maximum torque of 0.14 foot-pound needed in this example. Any usefulness which the magnetic-bar control might have would be restricted to cases where very small torques would be required.

Disturbances

Some of the disturbances which a satellite is subject to are considered. The first of these is the moment due to the gravitational gradient. A sphere with uniform density, as the satellite was first assumed to be, would have no gravity moment. In order to visualize the magnitude that the gravity moment could have, assume that the satellite is shaped to have the maximum gravity moment. Such a configuration would have all the weight condensed into two spheres. It is assumed that these spheres are separated 6.6 feet so that the moment of inertia is still 1,000 slug-feet^2. For this simplified configuration the gravity moment is given by the expression

$$\text{Moment} = \frac{6c^2We^2\sin \phi \cos \phi}{r^3}$$

where

c radius of the earth

r radius from center of earth

ϕ angle from vertical

W weight at sea level of one sphere

e semi-distance between spheres

The maximum moment occurs when the satellite is 45° to vertical and is 0.00187 foot-pound. A more detailed discussion on the method of determining the gravity moment is given in reference 3.

Another disturbance which the satellite may be subject to is one arising from radiation pressure from the sun. This pressure is of the order of 1×10^{-9} pounds per square inch (ref. 4 or 5). Therefore, if a 10-foot-diameter sphere was painted so that one-half was nonreflecting and the other half reflecting, the maximum moment due to the radiation pressure would be of the order of 6×10^{-5} foot-pounds. A similar calculation is made for the Vanguard vehicle in reference 5.

It can be seen that these disturbing torques are small compared with the maximum torque of 0.14 foot-pound required to perform the test maneuver. Therefore, the control system would have no difficulty in overcoming these disturbance torques. The disturbing torques, however, could become a problem if they caused a continuous out-of-trim condition. Such a condition would exist if, for example, the principal axis of the satellite was displaced from the control-line axis and this displacement caused a net out-of-trim gravity torque. This condition would require a continuous flow of fuel with the jet system or would cause the inertia-wheel system to exceed its maximum angular velocity.

This difficulty could be avoided if some means for providing a trim moment exists. One means of providing such a trim force would be the use of permanent magnets. As was pointed out before, interaction with the magnetic field of the earth can provide rolling and yawing moments on an equatorial orbit and pitching moments on a polar orbit. Also, the gravity moment will provide pitching and rolling trim moments. As regards the maneuver used in this study, if the principal axis is alined with the control line, the gravity moment would provide useful trim torques. Thus, with the exception of yawing moments in polar orbits, a combination of these two factors will provide all the needed trim moments.

COMPARISON

The total weight for each system for any number of cycles can be computed. The results are shown in figure 4 as the total weight for 0 to 1,000 cycles. It can be seen that the weight of fuel is the predominant factor in the weight of the jet system for a large number of cycles. The inertia-wheel system with solar batteries offers a great weight-saving advantage for large numbers of cycles, whereas the jet system weighs less for a limited number of cycles. It should also be noted that a control system for one degree of freedom can be devised which weighs less than 1 percent of the weight of the satellite.

The effect of using a different time history for the change in attitude angle is now considered. Consider an alternate maneuver in which a relatively high angular acceleration, as compared with the initial tracking acceleration, is held until a certain angular velocity is reached. This angular velocity is such that it will result in an attitude change of 180° in 400 seconds, which is the same attitude change that was achieved in the tracking maneuver. This alternate maneuver will not satisfy the tracking requirements of the initial maneuver but is given as an example of a different type of maneuver. The time histories of the two maneuvers are plotted in figure 5. The alternate maneuver is well suited for a jet system. The lower angular velocity required means that less fuel weight is required. The energy required of an inertia-wheel system to perform the

alternate maneuver is much less than that for the tracking maneuver. However, the long period of time that the flywheel would have to be kept at maximum angular velocity, and the resulting friction and heat losses, detracts from the energy-saving advantage of this type of maneuver for the inertia-wheel system. Also, the higher angular acceleration required means that the torque and power outputs of the motor would be increased, and thus a larger motor would be required.

CONCLUSIONS

On the basis of an assumed tracking task that involves relatively high power and energy requirements, a comparison has been made of the weight of a jet-reaction control, an inertia-wheel control, and a magnetic bar that uses interaction with the magnetic field of the earth for controlling the attitude of a satellite. This comparison results in the following conclusions:

1. A control system for one degree of freedom can be devised that weighs less than 1 percent of the weight of the satellite.

2. The inertia-wheel system with solar batteries offers weight-saving possibilities if a large number of cycles of operation are required. The jet-reaction system would be preferred if a limited number of cycles were to be performed. In general, the number of cycles at which the weight of the jet control will equal the weight of the inertia-wheel control will depend on the details of the maneuver to be performed. The magnetic-bar control requires such a large magnet that it is impractical for the example application but might be of value for supplying small trimming moments about certain axes.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Field, Va., October 1, 1958.

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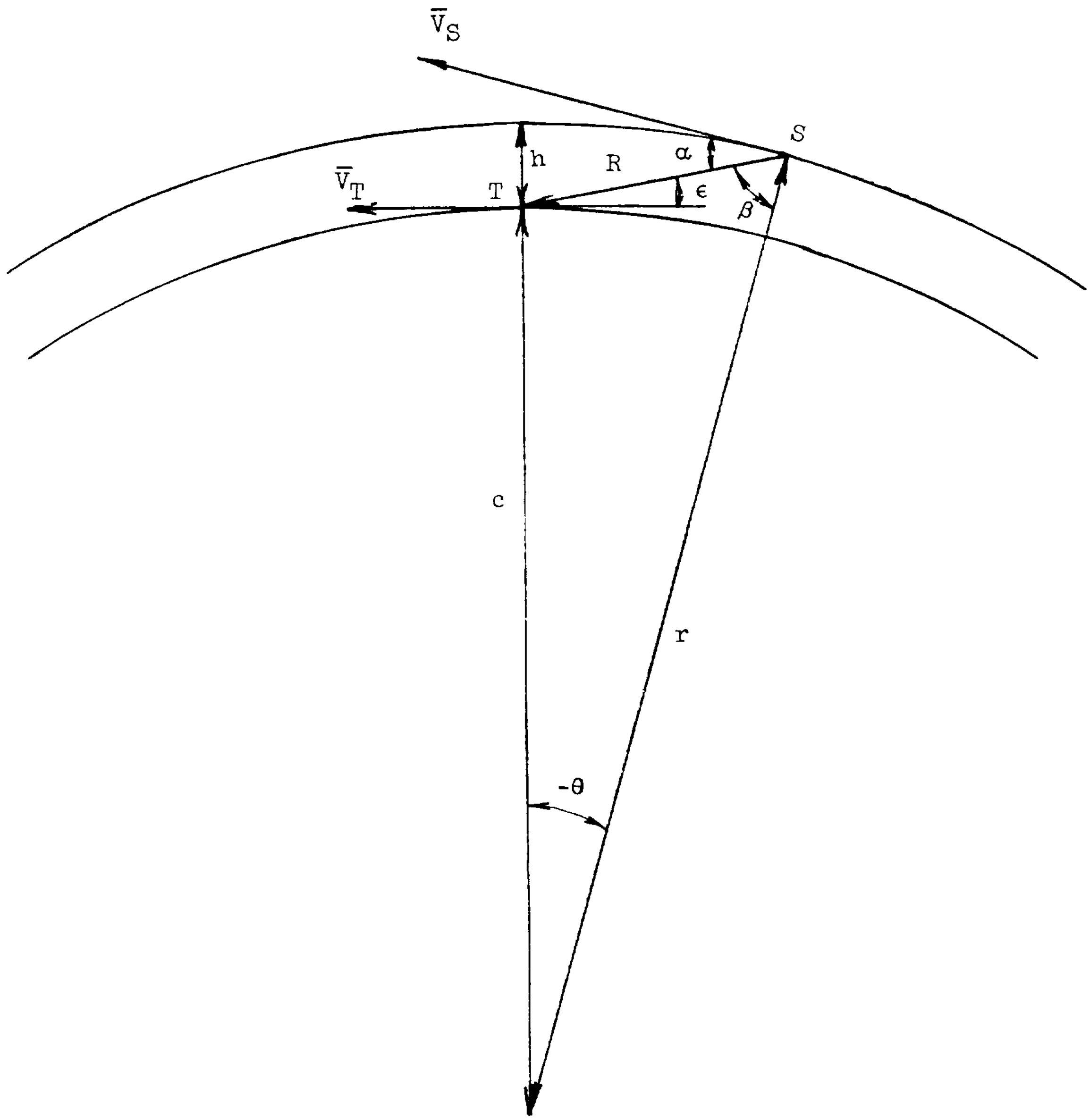

Figure 1.- Geometry of the tracking problem.

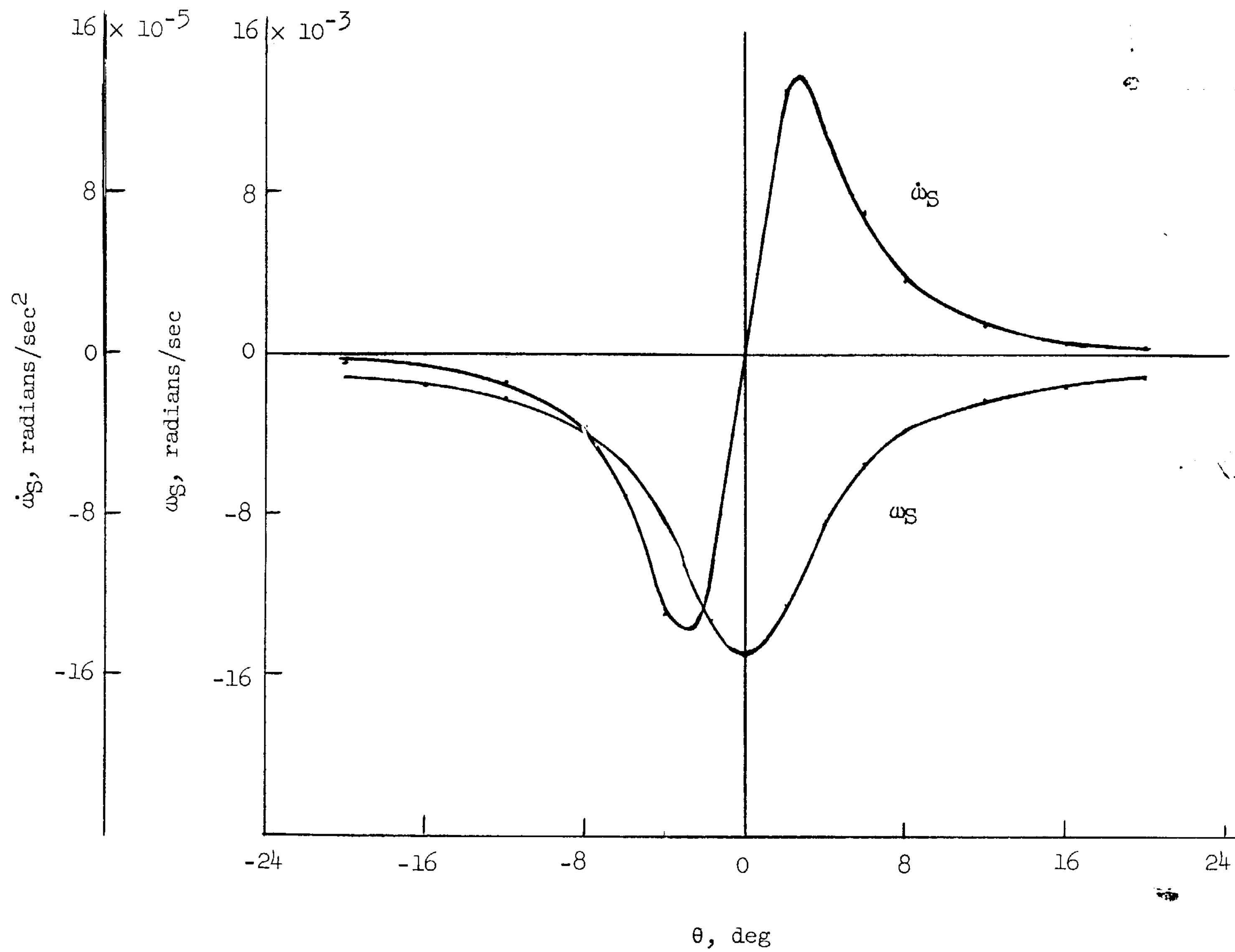

Figure 2.- Angular velocity and angular acceleration for tracking task.

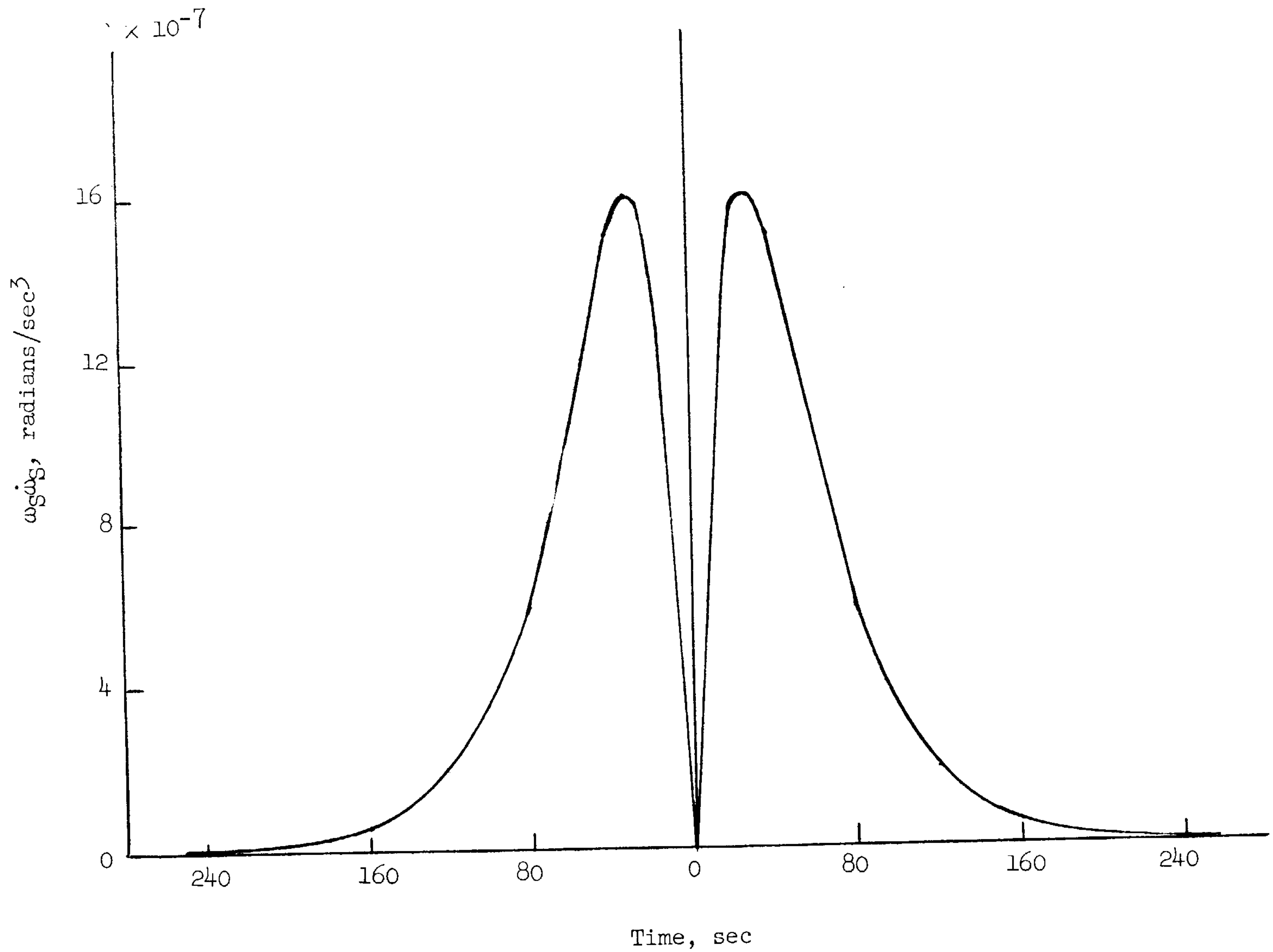

Figure 3.- Variation of the product of ω_S and $\dot{\omega}_S$ with time.

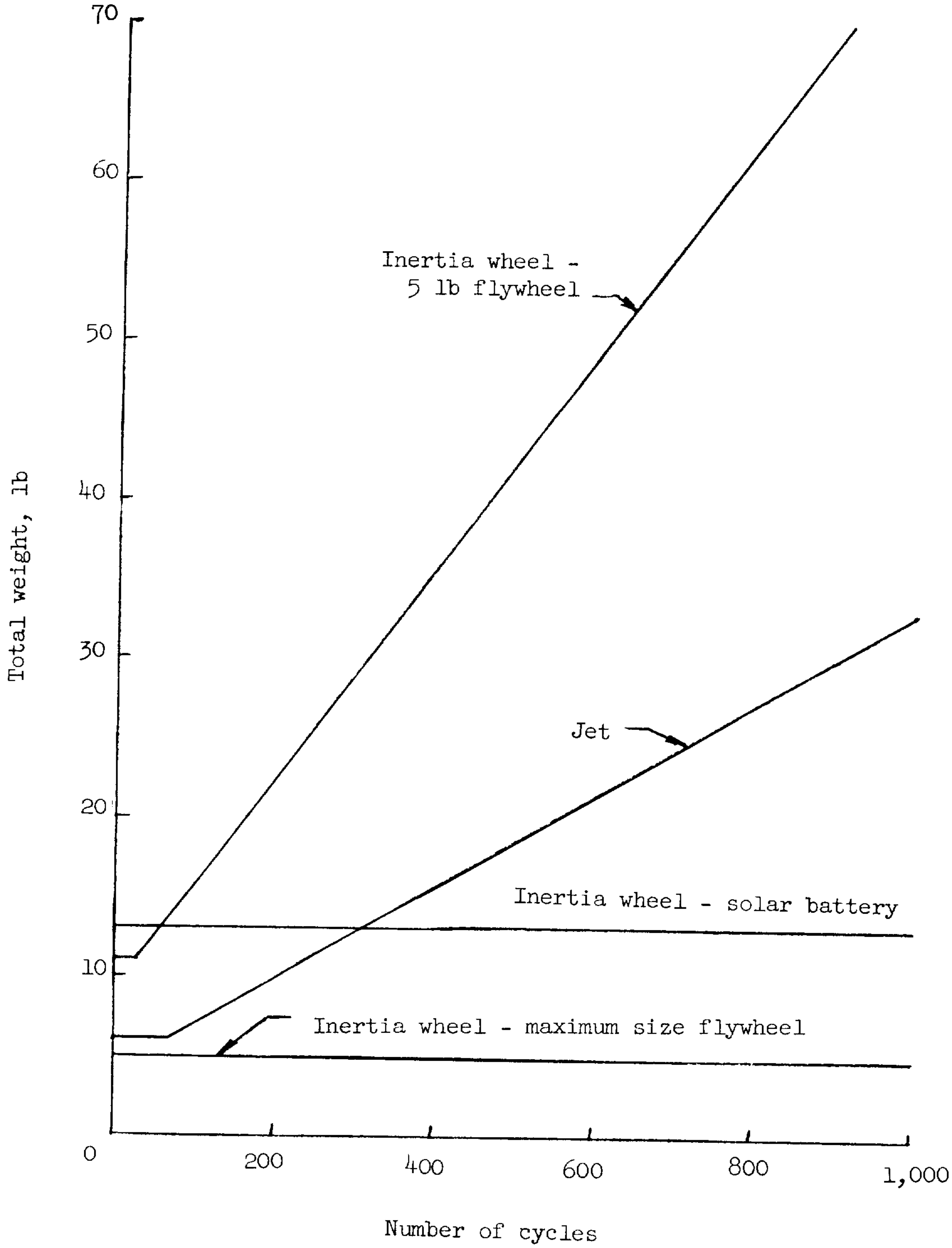

Figure 4.- Variation of the total weight of various control systems with number of cycles of the tracking maneuver.

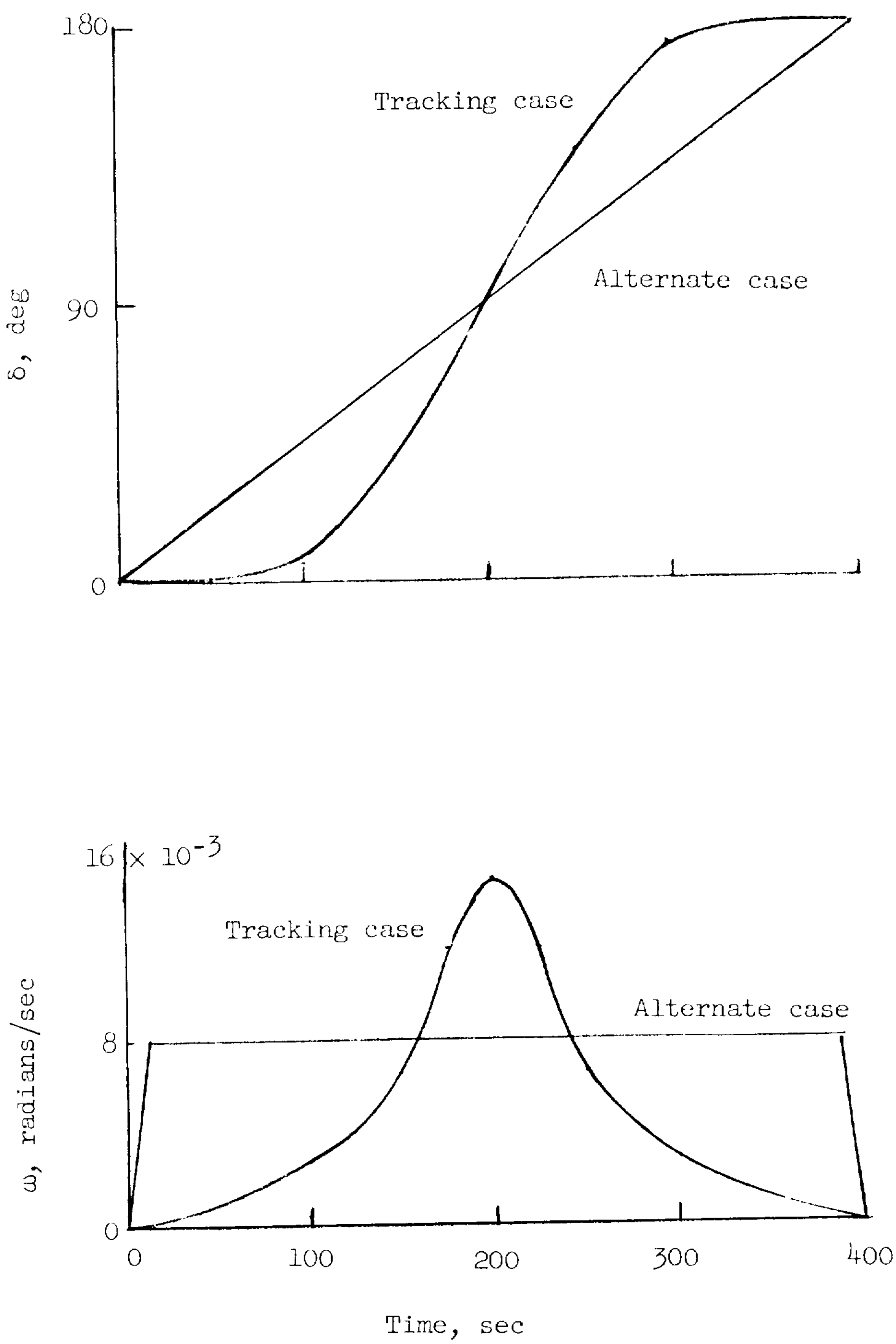

Figure 5.- Time histories for tracking maneuver and alternate maneuver.

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